loan copy 6/11/63

1/8 p

CR52198

~~X63 16889~~

N65-88906

code 2A

2

(NASA CR-52198)

CONTRIBUTION TO AGARDOGRAPH ON

SPACE SIMULATION CHAMBERS AND TECHNIQUES

By Keirn Zebb

Jet Propulsion ~~Laboratory~~

~~California~~ Institute of ~~Technology~~

Pasadena, ~~California~~

April 29, 1963 ref 2 Submitted for Publication

CONTENTS

I. SOLAR RADIATION SIMULATION

A. INTRODUCTION

Achievement of the high reliability and long life which are essential to successful spacecraft requires careful design and thorough, well-planned testing programs. An important part of such testing, required for both prototype qualification and flight-unit acceptance, involves the exposure of complete and operating spacecraft to simulated space environments. Vacuum, cold-space background (heat sink), solar radiation, planet albedo and thermal radiations, shock, vibration, meteoroids, reduced gravity, etc., all constitute important elements of the space-flight environment. It is not feasible at the present time to construct test facilities in which all of these influences can be simulated simultaneously, but various studies have shown that it is practical (as well as necessary) to include vacuum, heat sink, solar radiation, planet-related radiations, and possibly one or two others.

This chapter is devoted entirely to the problems of simulating solar radiation, and it describes some of the equipment and techniques which are being used or considered for this purpose. It must be appreciated at the outset, however, that the state-of-the-art in solar simulation techniques is only now evolving--this in spite of the fact that several large facilities are nearing completion. Meaningful experience and demonstrated performance are still extremely limited making detailed assessments of equipment and methods premature. Furthermore, many of the systems now being developed have proprietary aspects, and details concerning them are, therefore, often not available.

In view of these matters, the approach here will be to try to indicate what needs to be accomplished by solar simulation, and then to illustrate the complexity of the problem and the diversity of proposed solutions by describing some specific configurations and equipment, rather than to try to formulate general design information.

B. SPACECRAFT THERMAL CONTROL TESTING

Experimental investigations of the heat balance and temperature distributions of spacecraft constitute some of the most important applications of space simulation facilities. This is because engineering calculations of these characteristics, for all except the simplest examples and configurations, are extremely difficult to formulate and carry out. Such calculations must account for intricate internal power distribution systems, and be able to handle a large number of complex conductive and radiative interactions between various spacecraft subsystems and components. Furthermore, necessary data such as the thermal radiation properties of materials and coatings or joint conductances are often unavailable or inaccurate in practical applications, and they can only be determined or verified by experimental procedures.

At various stages of a thermal design tests may be carried out on assemblies or portions of a spacecraft, but the complexity of the interactions is so great that it is usually regarded as necessary ultimately to place the entire spacecraft in a chamber. This requirement combined with overall systems test procedures, establishes the need for at least some relatively large test facilities. Perhaps these remarks can be more fully appreciated if one visualizes a typical spacecraft configuration such as the Mariner shown in Figure 1.

Techniques using scale models have not been developed, nor do they appear to offer much promise because of the complex nature of the heat transfer paths within a spacecraft and because of the necessity for representing on-board energy sources.

Two basic facility concepts and test techniques have been developed. In both of these, the test is conducted in a vacuum chamber to eliminate conductive and convective heat transfer and to assure radiative exchanges only. For simple performance testing, it is only necessary to produce the predicted temperatures in the spacecraft. This can be accomplished by the use of heaters of various types combined with control of the chamber wall temperature, typically in the range between -65°C and 100°C. However, for configuration development and thermal design studies the solar flux must be simulated, and the walls must be cooled to a sufficiently low temperature (typically to about 100°K by means of liquid nitrogen) to simulate the cold-space background.

Here we are concerned principally with the second of the testing techniques mentioned above, and in particular with the equipment and methods used in providing simulation of the solar thermal radiation.

C. CHARACTERISTICS OF SOLAR RADIATION

The solar energy flux incident on a spacecraft can be described by its intensity, spectral distribution, uniformity, and collimation. Obviously, because of the great distances involved, the solar flux is completely uniform as viewed by any conceivable spacecraft. The collimation also is nearly perfect, it being defined by the fact that at Earth's orbital distance, the sun subtends an angle of only 32 minutes. The corresponding figures for Mars and Venus are 20 minutes and 46 minutes, respectively.

The bulk of the energy in the solar spectrum lies between the wavelength limits of 0.28 and 5.0 microns with only about 0.5% of the energy lying beyond each of these limits. Approximately 9% of the total energy is in the ultra-violet at wavelengths shorter than 0.40 microns; 46% is in the visible range from 0.40 to 0.76 microns, and 45% is in the infra-red range beyond 0.76 microns. Typical spectral distribution data for conditions outside the atmosphere are plotted in Figure 2, and detailed information and tabular data are available in References 1 and 2.

As has often been pointed out, the visible and infra-red portions of the solar spectrum are well approximated by the radiation from a black body at about $6,000^{\circ}$K.

Solar radiation at the Earth's surface differs from that outside the atmosphere in that the ultra-violet and some portions of the infra-red are largely removed, and the total intensity is reduced to about 75% of its incident value by atmospheric absorption.

The total energy falling on a unit area is equal to the area under the spectral distribution curve (Figure 2) and has the value 0.140 watts/cm^2 (130 watts/ft^2) at the mean distance of the Earth from the sun. The intensity varies with distance from the sun in accordance with the inverse square law. A spacecraft near the orbit of Mars would experience an intensity of about one-half this value, and one near the orbit of Venus would find the intensity nearly twice as great as at Earth.

D. DESIGN CRITERIA FOR SOLAR SIMULATION SYSTEMS

At the present time the design and construction of large-area solar simulation systems is a difficult task. As this is written, actual operating experience with large systems is still extremely limited, and many of the important design concepts being considered have not yet been fully demonstrated. The state-of-the-art does not enable designs satisfying all desirable criteria, and a number of compromises involving tradeoffs among uniformity, collimation, intensity, and spectral distribution are usually required. For example, test applications involving solar energy concentrators would stress collimation over uniformity, whereas in tests of large assemblages of solar cells, just the reverse would be true. Collimation is important in heat balance and temperature distribution tests because it affects the locations and intensity of shadows on the spacecraft, determines the amount of energy incident on surfaces which are intended to be aligned with the sun's rays, and influences the reflection patterns between spacecraft parts. Such effects are probably especially important for spacecraft types which are characterized by open structures and appendages, as contrasted with more or less enclosed bodies. In some instances one of the most important compromises may involve improved collimation with a reduction in intensity.

The spectral distribution of the incident energy is important because many spacecraft materials are spectrally selective, having absorptivities which are strong functions of the wavelength. This matter is discussed in more detail later in connection with the description of available energy sources.

It is not reasonable at this time to set down definitive criteria or specifications for solar simulation systems in general because, as already remarked, these would not be the same for all intended applications. Furthermore, the performance attainable should improve during the next few years. It will, however, help to fix ideas to state briefly some of the general objectives which are being sought or which are considered as attainable at the present time.

Intensity: For Earth orbiters and lunar spacecraft the intensity requirement is 130 watts/ft^2. For interplanetary spacecraft the intensity should be variable, the requirements at Mars and Venus being, respectively, approximately half and twice the value above at Earth.

Uniformity: ±5% to ±10% appears to be attainable, measured with a detector having a sensitive area of about 1 foot square. It is desirable to achieve ±5% uniformity to a detector of only 2 or 3 inches on a side.

Collimation: Some systems have as an objective about ±1 degree, and some users feel that they need this or better. However, others consider ±2 or 3 degrees as acceptable and more realistic of attainment.

<u>Spectrum</u>: Specifications of this feature generally are not precise because once the energy source and configuration is selected, the spectrum will be whatever it turns out to be unless filtering is employed. Desirable specifications would require energy content in 0.1 micron wavelength intervals to be within ±5% to ±10% of that in the corresponding interval of the solar spectrum throughout the range from about 0.3 to 1.5 microns. Greater deviations would be permissible at longer wavelengths.

Simulator size and expected test durations are other matters which may have important influences on the design of a system. On the question of size, it can be remarked that applications vary widely from laboratory devices irradiating only a few square inches to large environmental chambers perhaps a hundred feet or more in diameter. A number of the present generation of large chambers turn out to be about 25 to 35 feet in diameter with planned areas of irradiation about 20 feet in diameter.

Test durations, that is periods of continuous simulator operation, are expected to vary from a few hours to many days for qualification and life tests of flight spacecraft.

B. DESIGN OF SOLAR SIMULATION SYSTEMS

1. Examples of Proposed Configurations

Since standardized designs and design procedures have not yet appeared in the solar simulation field, the present state-of-the-art is best described by examples of systems being built or proposed for current space simulator projects. These will illistrate the complexity of some of the problems involved and demonstrate the nature of some of the solutions being attempted. It should be appreciated that most of the systems under development still have proprietary aspects, and some details about them are, therefore, not available.

The first example chosen is the original* design for the 25-Foot Space Simulator at the Jet Propulsion Laboratory, Pasadena, California. This system illustrates a number of important points, and is the one to which the writer had access for the greatest amount of detail. The JPL facility and the solar simulation system are shown schematically in Figures 3 and 4. Photographs of several details appear in Figures 5 through 9, inclusive. The configuration consists basically of two Cassegrainian mirror systems placed back to back with a quartz transfer lens between them. The radiation source consists of an array of 131 2.5-KW mercury-xenon compact-arc lamps (Figure 6). Each lamp is housed in its own 16-inch diameter glass reflector assembly (Figure 5), which collects the radiation and directs it downward. The upper mirror, rather than being a continuous surface, is actually a group of 19 32-inch diameter flat stainless steel

---

* Recent modifications have been completed and will be described later.

mirrors. Each receives the light from a cluster of seven lamps. The flats direct the light upward to the pseudo-hyperbola (Figure 7), and this assembly is imaged by the 3-foot diameter quartz transfer lens onto the virtual source inside the vacuum chamber.

The virtual source assembly (Figure 8) is about 30 inches in diameter, and is covered with some 1,150 small convex reflectors, or "pebbles" as they are called. Each small reflector is shaped to reflect the light which falls on it to all parts of the inside parabolic reflector. This feature of the virtual source results in a uniform test beam even though the beam incoming through the lens is not uniform. The "pebbles" are shaped also to compensate for the intensity fall-off which occurs toward the outer portions of parabolic reflectors. The shadow of the virtual source which would occur on the axis of the test beam is filled by allowing some light to pass through a central hole and then reflecting it in a secondary system similar to the primary virtual source and paraboloid.

The 25-foot diameter parabolic mirror on the inside of the chamber (Figure 9) is a mosaic composed of 324 pieces. Each piece is supported and adjusted by three rods and is cooled by conduction through braided copper cables to cold gaseous nitrogen piping.

This system was designed and constructed by a contractor to a simple functional specification. It was originally intended to furnish approximately Venus intensity (275 watts/ft^2) to a 15-foot diameter test zone with a uniformity of ±10% and a collimation of ±5.3 degrees for the extreme rays with half of the energy within

±3 degrees. However, during the design, it was found that these performance characteristics could not be achieved within the constraints of cost, schedule, etc., and they were modified to 200 watts/ ft^2 on an 11-foot diameter. Unfortunately, the system as built, in spite of its apparent sophistication, did not achieve even the reduced objectives in terms of uniformity and intensity. The configuration has since been modified on a temporary basis by JPL to concentrate the available energy into a 5-foot diameter test beam. This modification required redesign of the virtual source to direct the energy to a smaller region of the collimating parabola. At the same time the virtual source was tipped a few degrees thus producing an off-axis test beam and avoiding the shadow under the virtual source. Intensity surveys through the resulting beam are shown in Figure 10.

A quite different arrangement appears in Figure 11, which shows the system planned (Reference 3) for the General Electric Company's Space Environment Simulation Laboratory at Valley Forge, Pennsylvania. This simulator is intended to furnish Earth-orbit intensity (130 watts/ ft^2) in a 20-foot diameter beam with a uniformity of ±5% and a collimation of ±1.5 degrees.

The energy is supplied by 148 5-KW xenon arc lamps clustered into four groups as shown. Reflective and refractive optics are used to collect the energy and direct it through the chamber wall to illuminate the four off-axis parabolic sectors in the dome of the chamber. The light is then reflected downward into the test zone.

The lamp reflectors for this system are rather complex, multi-zoned ellipsoids. This complexity is required to fit the reflector to the finite sized arc, and to furnish a suitably non-uniform beam to the collimating mirrors so that upon reflection, a beam of uniform intensity will result.

A similarity will be noticed in the JPL and GE systems in that in both systems the energy from many lamps is collected and superimposed to form a relatively large and concentrated source. The energy is then spread again by collimating reflectors to irradiate a large area. By contrast, some systems employ a multi-modular concept in which each lamp illuminates a small zone directly beneath or opposite it in the test volume. The systems shown in Figures 12-14 are all of this latter type.

Figure 12 shows a schematic drawing of the solar simulator module being developed for the Goddard Space Flight Center at Greenbelt, Maryland. These modules incorporate 2.5-KW mercury-xenon arc lamps, and the complete facility will employ 127 such units. The performance objectives of this system are as follows:

<u>Intensity</u>: Variable from 50 watts per square foot to 130 watts per square foot.

<u>Uniformity</u>: ±10% over any one square foot.

<u>Collimation</u>: ±4 degrees half angle.

<u>Test zone size</u>: 20 foot diameter circle overall with the conditions stated above in a 17 foot diameter, 20 feet high, cylindrical volume.

This module has been designed to accept a 5-KW lamp also; this change would be expected to approximately double the maximum intensity available. It has been found that the ellipsoidal reflector must be made and located very accurately for satisfactory results. It is expected that these will be manufactured as thin-film epoxy replications. The condenser and relay lenses are made of the best optical grade of quartz. The hyperbola also is made of quartz and of course has a reflective coating. The parabolic collimating reflectors are electroformed.

A solar simulator module utilizing a carbon-arc as an energy source is shown in Figure 13 (Reference 4). This module has been developed for use in the Mark I Aerospace Environment Simulator which is being constructed at the U. S. Air Force Arnold Engineering Development Center, Tullahoma, Tennessee. The module produces a uniform, collimated beam of Earth orbit intensity (130 watts/ft^2) in the form of a hexagon measuring 24 inches across the corners. In the Mark I facility these modules will be arranged to direct their energy through the sidewall of the chamber. Initially 77 modules will be nested to irradiate an area 5 feet wide by 32 feet high.

As can be seen in the figure the optics are rather complex and combine both reflective and refractive elements. Approximately one-quarter of the energy is transmitted through the central refractive elements, and the remainder is collected by the reflective system. A ring-shaped mercury-arc tube, some 4 inches in diameter, is mounted just in front of the field lens. The purpose of this lamp is to provide supplementary ultra-violet energy since the carbon-arc radiation is deficient in this spectral range. The highest quality optical components are used throughout. Two sapphire lenses are used along with quartz in the refractive system to reduce chromatic aberration and pass the wide spectrum. The entrance window to the vacuum chamber is also of sapphire, $\frac{1}{4}$-inch thick and 4-inches in diameter.

The performance objectives for this system include intensity uniformity of ±5% and collimation of ±0.75 degrees.

A carbon-arc unit capable of unattended operation for a 24-hour period has been designed for use in the module (Reference 5). This requires a mechanism to automatically join and feed positive carbons to the arc. The negative electrode is of sufficient length to give one day of operation before replacement.

Still another configuration for a solar simulation system is shown schematically in Figure 14. The concepts suggested here, together with other features of the energy collection and projection systems, are under development by Space Technology Laboratories, Inc., Redondo Beach, California. Like two of the systems described previously, the STL configuration involves a modular design, and like the Mark I design it employs carbon-arcs as energy sources. An important difference, however, is that the STL design provides off-axis collimators, whereas the other two modular systems described employ on-axis optics. The performance objectives of the STL system include intensity uniformity of ±10% and collimation to ±1 degree.

2. Discussion

The selection of solar simulation configurations described above is not intended as all inclusive, but rather to illustrate and emphasize the wide variety of concepts which are under consideration. The examples suggest various ways in which optical systems for solar simulators may be classified:

a. Systems may use principally reflective or refractive optical elements.

b. Sources may be either on-axis (JPL, GSFC, Mark I) or off-axis (GE, STL).

c. Large systems can be assembled either by collecting the energy from many lamps and superimposing it to form effectively larger sources (such as JPL and GE), or multimodular arrangements are possible in which each individual lamp illuminates its own zone (as in the GSFC, Mark I, and STL configurations.)

How these various possibilities are combined will depend on a number of factors, including the following:

a. The overall size of the system to be built.

b. The intensity, uniformity, and collimation required in the radiation beam.

c. The type of lamp or other energy source which is selected.

d. The costs involved and the construction schedule.

e. The state of development of needed components.

A few brief comments will suggest some of the considerations which arise:

Refractive optical elements must be made of the highest grades of quartz in order to withstand heat loads and to retain a reasonable degree of energy transmission throughout the spectral range. Large refractive elements or numerous elements for large systems become very expensive. Furthermore, they usually introduce difficult sealing and cooling problems. Reflective elements, on the other hand, are relatively inexpensive, and they are not particularly limited as to size. This is not to imply, however, that accurate shapes, high reflectivity, and durability are easily achieved.

Off-axis systems have been proposed in both large sizes and in multi-modular arrangements. They have an advantage over on-axis systems in avoiding the shadow created by the source in the center of the beam. A further very important point favoring off-axis systems is that they minimize the problems associated with the return of energy to the test item which was previously radiated or reflected by it.

Difficulties with radiation continuity and uniformity can be expected at the edges of adjoining beams, and of course, the opportunities for such difficulties to appear are more numerous with multi-modular systems than with those that collect and then spread the energy from many lamps. Systems of the latter type have a further advantage over the multi-modular types in that variations in lamp characteristics from lamp to lamp, or even complete failures of lamps, have only a small effect on the intensity and uniformity of the ultimate beam. In addition, intensity in the test beam can be controlled simply by selecting the proper number of lamps. In favor of multi-modular arrangements, on the other hand, is the fact that only as many modules as required to cover the test item need be used, thus saving wear and tear on unneeded units, and minimizing the heat load on the cryogenic systems.

These questions of the advantages and disadvantages of the various methods of solar simulation which have been undertaken could be examined in considerably greater detail, but it is probably not profitable here to do so. There are nearly as many opinions about the best system as there are simulator projects and people involved. However, until some of the systems have been completed, and actual operating experience has begun to accumulate, many of the supposed advantages and disadvantages of different systems will remain somewhat matters of conjecture.

F. ENERGY SOURCES FOR SOLAR SIMULATION SYSTEMS

At the present time the greatest obstacle to the design of efficient and effective solar simulation systems is the unavailability of suitable energy sources. The only practically available sources are the carbon-arc and the xenon and mercury-xenon compact-arc lamps. All of these lamps were developed originally for signaling, searchlight, and movie projection purposes. Although they have undergone significant development during the past year or two, they still do not provide as compact and intense sources as are needed. Other types of high-power arcs, plasma jets, and even pools of molten metal, have been considered as possible sources, but the packaging, control, and practical application of these have not been demonstrated.

1. Descriptions of Arc Lamps

Carbon-arc lamps are available in several sizes up to about 12KW. The larger sizes cost $5,000-$8,000 each including the essential accessories. A reasonable estimate is that 30-35% of the electrical input appears as radiation and that of this perhaps 75% can be collected as the input to a solar simulation system. The carbon-arc lamps suffer from the disadvantage that they are large and cumbersome to handle and operate, and provision must be made to remove, by blowing or suction, the dust and ash which result from the burning and which would otherwise collect on the optical surfaces. Electrode burning times are limited to about 20 minutes in some models and up to an hour in others. This difficulty is being overcome by the development of automatic electrode changing and feed mechanisms which will provide up to 24 hours of continuous lamp operation without attention. In spite of these difficulties the carbon arcs are attractive sources because, as will be discussed later, they offer an excellent spectral distribution.

The compact-arc lamps (Figure 6 and References 6 and 7) consist of a quartz envelope containing xenon at a pressure of several atmospheres and mercury also in some models. The arc discharge takes place between tungsten electrodes which are separated by several millimeters, the exact dimension depending on the design and power rating of the lamp. These lamps are becoming available in a large variety of models from several manufacturers. For solar simulation applications the models with power ratings from about 2 to 5 KW are the most useful at the present time. Experimental lamps up to 10KW have been built and could become of considerable interest if the arc and bulb dimensions can be held sufficiently small. The compact-arc lamps are relatively efficient in that approximately 50% of the electrical input is radiated in the spectral range from 0.2 to 1.4 microns with additional energy at longer wavelengths.

Practical experience is insufficient as yet to establish the life of these lamps in solar simulator applications. The bulbs age by darkening of the quartz envelope, caused principally by deposition of electrode material. A decrease in useful output of about 25% defines a reasonable life, and it is estimated that perhaps 500 to 1000 hours of operation can eventually be realized. Replacement lamps cost in the range of $300-850 depending on type and power rating.

The number of lamps required and the power consumed by a large simulator are considerable. Overall efficiencies (that is, the ratio of useful radiant power delivered to the test zone to the electrical power input) will, judging from present designs and experience, apparently not exceed about 5%. This figure corresponds to somewhat more than one 2.5 KW lamp per square foot of test zone at Earth intensity.

2. Importance of the Radiation Spectrum

The spectrum of the energy delivered to the test zone is quite important. The equilibrium temperature of a spacecraft results from a balance between incoming and outgoing energy. The sources of incoming energy include nearby reflecting and emitting bodies, on-board power dissipation, and the energy absorbed from direct solar radiation. The outgoing energy is that radiated to "cold, black" space, which for practical purposes is at about 3°K. Neglecting for simplicity all inputs except the absorbed solar radiation, the energy balance is expressed, in very simplified form, by

$$\varepsilon \sigma A_s T^4 = \alpha E A_c$$

or

$$\sigma T^4 = \frac{\alpha}{\varepsilon} \frac{A_c}{A_s} E$$

where T = effective surface temperature of the spacecraft

E = intensity of the solar radiation

A_s = radiating surface area of the spacecraft

A_c = projected area of the spacecraft intercepting radiant energy

α = spacecraft surface absorptivity for solar radiation

ε = low temperature surface emissivity of the spacecraft

σ = Stefan-Boltzman constant

The purpose here is merely to show the dependence of the spacecraft temperature upon the ratio of the solar absorptivity to the low-temperature emissivity, that is upon the familiar quantity α/ε. This ratio may be varied by the choice of surface materials and treatments in a range from about 0.4 to 10, but the point of particular significance for the moment is that for many materials α is a strong function of the wavelength of the incident radiation. Consequently the effective or integrated value of α/ε depends upon the overall spectrum.

Typical spectral data for several sources are shown in Figures 15-17. These were prepared from information published in manufacturer's bulletins, and the conditions of measurement, as far as they are known, are noted on the figures themselves. Figure 15 shows the spectrum obtainable from a typical carbon-arc lamp, and Figures 16 and 17 show spectra for mercury-xenon and xenon compact-arc lamps, respectively. These distributions may be compared directly with the solar spectrum as shown in Figure 2. In each of these figures the lamp data have been scaled in such a way that the total energy represented (that is the area under the curve) up to 1.4 microns is same as that for the solar spectrum out to 1.4 microns (namely, 0.120 watts/cm^2).

Accuracy and consistency in data such as these are difficult to achieve. The measurements are difficult at best, involving elaborate instrumentation and techniques. Furthermore, the results depend on what part of the arc is viewed, what lenses and reflectors have been involved, how much electrode radiation is included, etc. Consequently, data obtained from different individuals or laboratories often differ somewhat.

Certain features, however, are characteristic. The mercury-xenon lamps are relatively too intense in the ultra-violet and visible range up to about 0.60 microns, and too weak in the range from about 0.60 to 1.0 micron. Xenon lamps, on the other hand, furnish considerably too much energy in the near infra-red range (0.8 to 1.1 microns) and are deficient in the ultra-violet and visible. Both the mercury-xenon and xenon spectra contain a number of intense lines which do not appear in the form of data presentation used here. For most materials these lines probably have no great significance in themselves as long as the energy they contain is included in the averaged intervals as shown.

The carbon-arc spectrum appears to be a continuum without many intense lines. Its characteristics depend in part upon the type of carbon electrode used and upon the composition of the electrode core material. The similarity of the energy distribution to that of the sun is evident.

It is important to recognize of course that lamp spectra are only the starting point as far as a solar simulator design is concerned. The optical-components in any system will be somewhat selective in their reflecting and transmitting properties as a function of wavelength, and the spectrum in the test area can only be estimated after taking account of these. For example, the reflectivity of aluminized surfaces is 0.95 or higher in much of the spectral range, but it falls to about 0.85 near 0.8 microns, which happens to coincide approximately with the peak in the xenon lamp spectrum. Hence, a xenon lamp spectrum is flattened somewhat after reflection from aluminized surfaces, and the effect can be quite significant if several reflections occur.

An important advantage of the energy collecting systems of the type described for the JPL and GE chambers can be mentioned here. It is that the spectrum in the final beam can, at least in principle, be controlled by mixing lamps of different characteristics. It has been suggested, for example, that a mixture of about two-thirds xenon and one-third mercury-xenon lamps would result in a better spectrum than obtainable with either lamp used alone.

One method of studying the spectral distribution problem is by evaluating the absorptivities of different materials of interest to the different spectra. This can be done either by direct measurements or by calculations which combine reflectance data as a function of wavelength with the chosen spectra. A few tentative calculated results for some common spacecraft surfaces have been collected in the following table:

Absorptivities of several materials
to different arc lamp spectra

Material	Absorptivity	Absorp. to lamp spectrum / Absorp. to solar spectrum			
		Carbon arc	HgXe	Xe	1/3 HgXe 2/3 Xe
Polished aluminum	0.235	0.97	0.99	1.00	1.00
Aluminum mirror	0.100	0.97	0.93	1.05	1.02
Gold	0.226	0.86	1.32	0.76	0.94
Aluminum silicon resin paint	0.247	0.98	0.98	1.02	0.99
Zinc oxide silicon paint	0.177	0.96	1.39	0.84	1.01

Further work is required, but some conclusions already appear to be indicated: For some materials the spectrum makes little difference, but for others differences from solar absorptivity as high as 30-40% may occur. The carbon-arc spectrum appears to be fairly acceptable for most materials. The expectation suggested by the table, that a mixture of mercury-xenon and xenon lamps could provide a better spectrum than either lamp used alone results from the fact that some materials having too high an absorptivity to the mercury-xenon spectrum have too low an absorptivity to the xenon spectrum.

The general solution of the spectral distribution problem is not easy. The possibilities include: (a) mixing sources in certain kinds of systems, (b) developing new lamps or other sources,

(c) using carbon-arc sources in spite of their operational disadvantages, and (d) filtering. The prospect of filtering is of course an unhappy one because it simply means throwing away energy in systems where the efficiency is already extremely low.

G. REFERENCES

1. Francis S. Johnson, Editor, "Satellite Environment Handbook", Stanford University Press, Stanford, California, 1961.

2. United States Air Force, "Handbook of Geophysics", Chapter 16, The MacMillan Company, New York, 1960.

3. A. Lengyel, P. A. Marfone and D. J. Santeler, "Design Criteria for a Space Environment Simulator", American Society of Mechanical Engineers, Paper Number 61-AV-52, March, 1961.

4. Charles F. Norman, "Solar Simulation Instrumentation", Arnold Engineering Development Center, Report No. AEDC-TDR-62-191, January, 1963.

5. Martin L. Levene and Carl Lauxen, "Continuous Carbon-Arc for Solar Simulation", American Society of Mechanical Engineers, Paper Number 62-WA-331, November, 1962.

6. Wolfgang E. Thouret and Herbert S. Strauss, "Design and Technology of High-Wattage Xenon and Mercury Compact Arc Lamps for Simulation of Solar Radiation", American Society of Mechanical Engineers, Paper Number 62-WA-215, November, 1962.

7. Otto E. Lienhard and John A. McInally, "New Compact Arc Lamps of High Power and High Brightness", Presented at National Technical Conference of the Illuminating Engineering Society, September 24-29, 1961, St. Louis.

H. LIST OF FIGURES

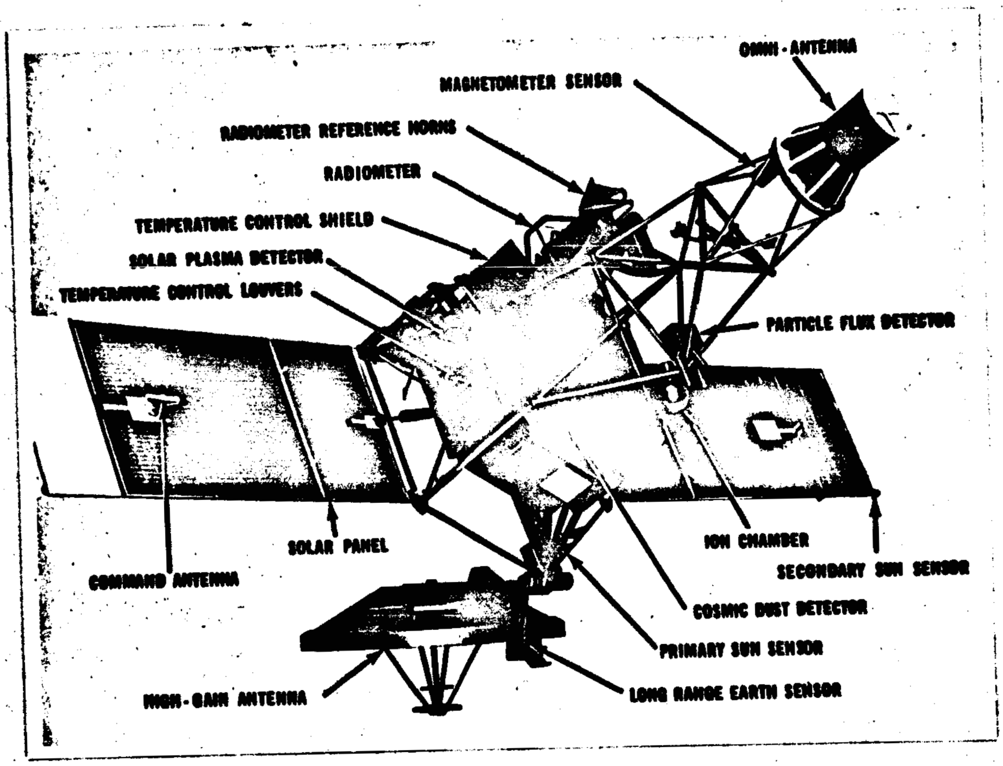


Fig. 1. Mariner 2 Spacecraft

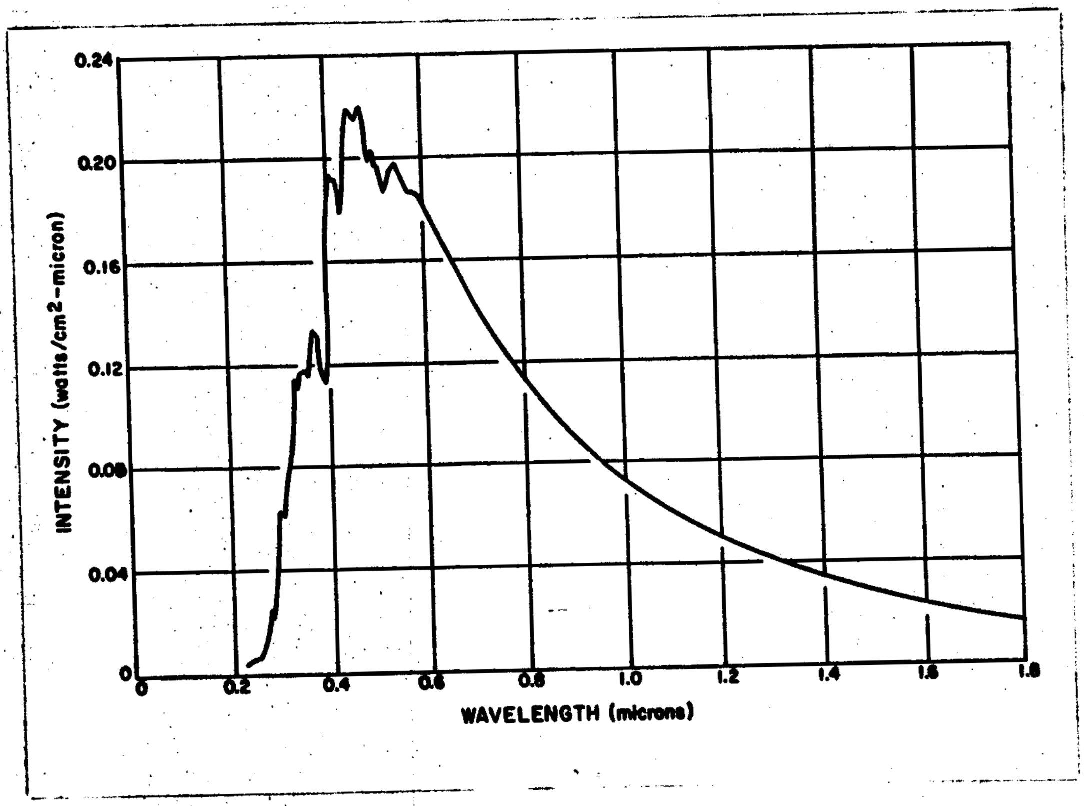


Fig. 2. Solar spectrum above the atmosphere at Earth's distance from the sun

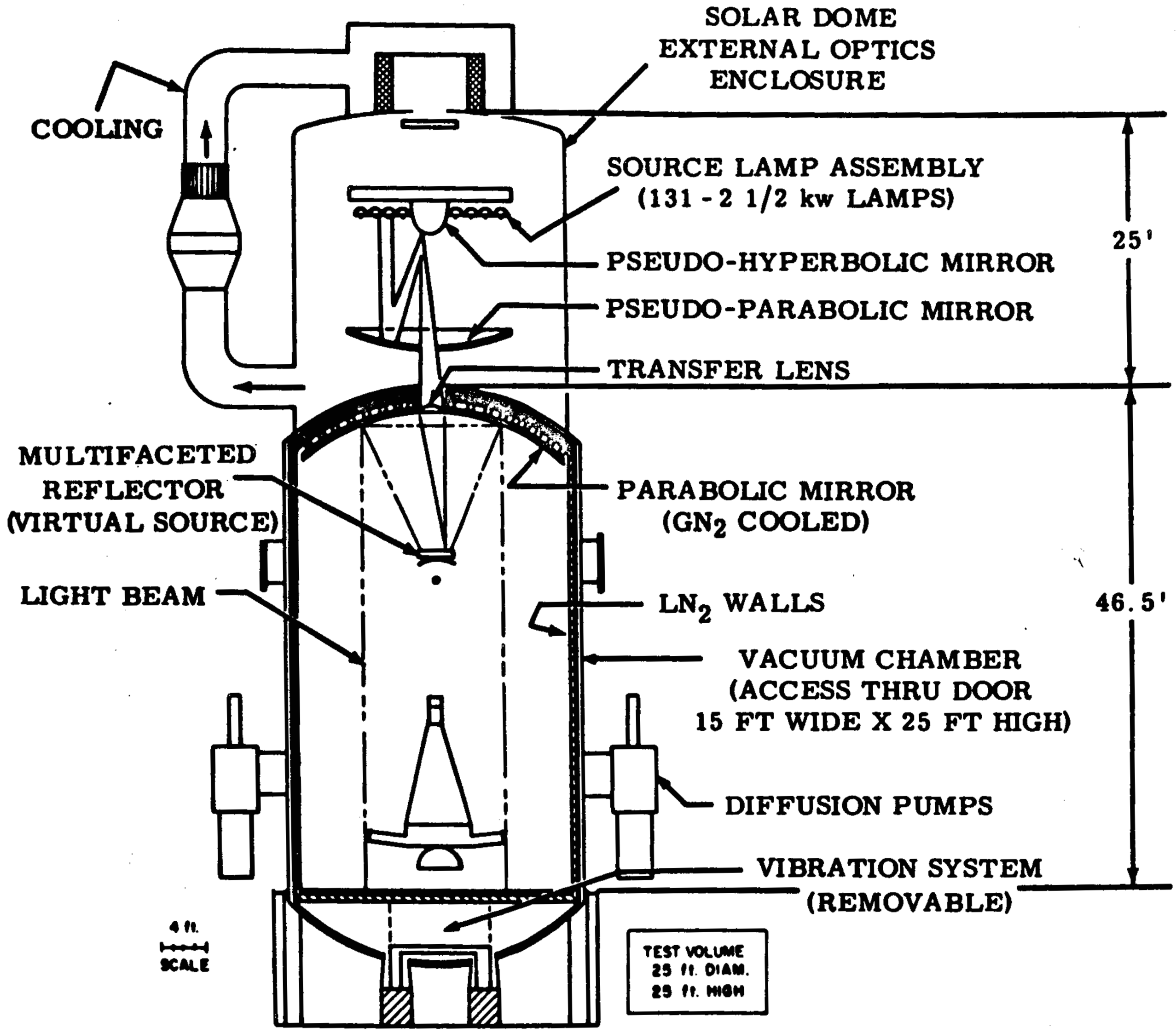


Fig. 3. Schematic drawing of the 25-Foot Space Simulator at the Jet Propulsion Laboratory

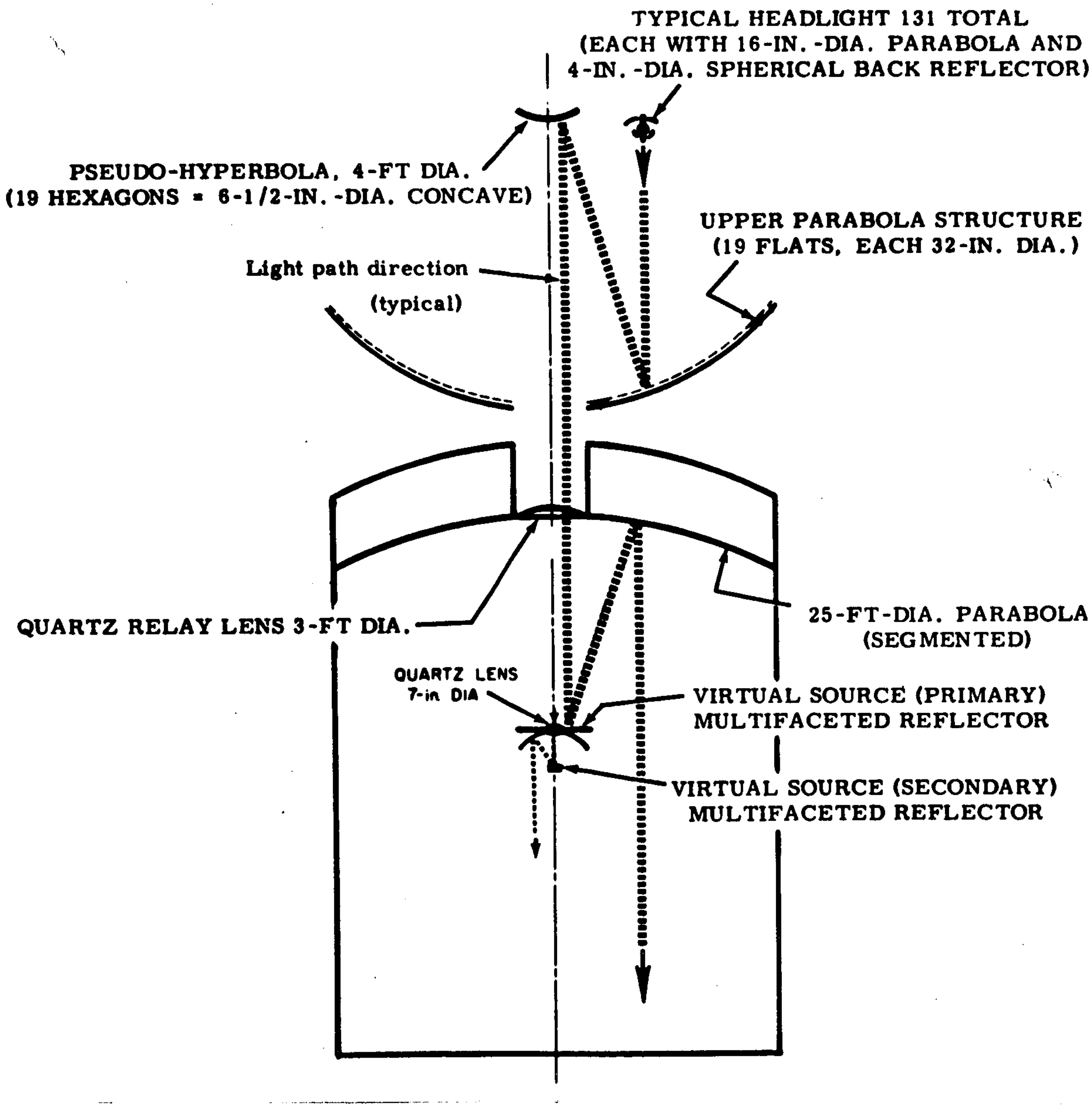


Fig. 4. Optical schematic for solar simulator (JPL)

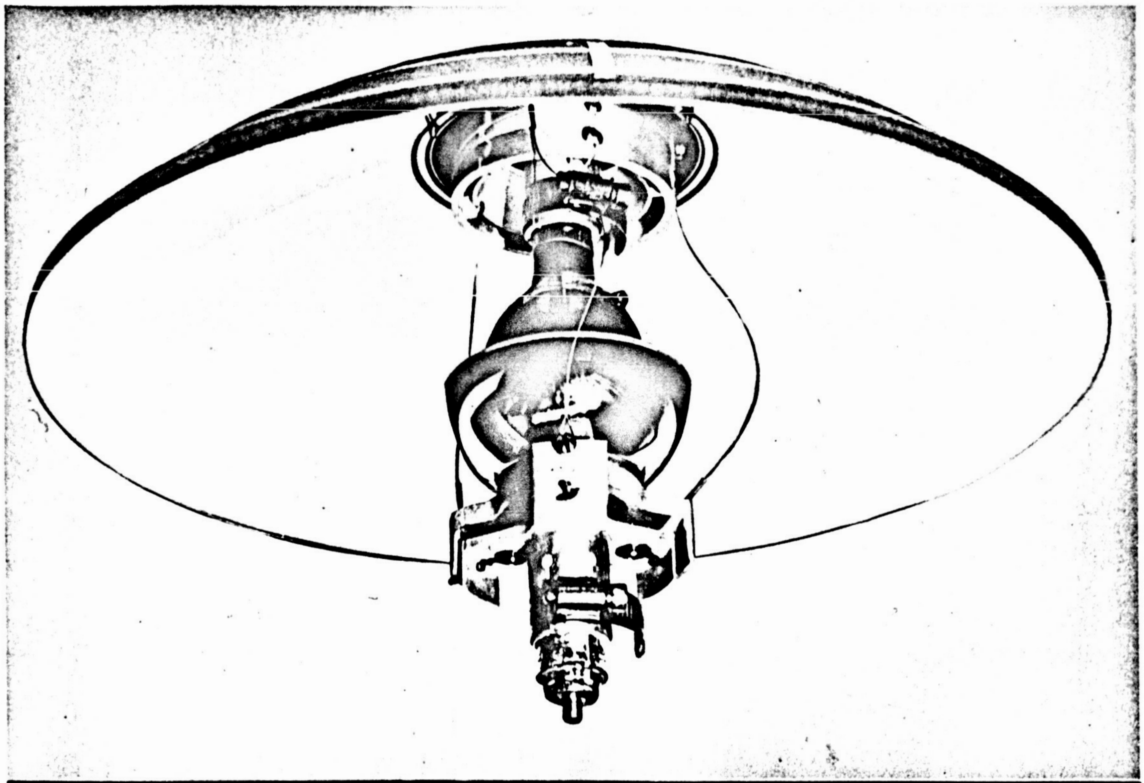


Fig. 5. Compact arc-lamp and reflector assembly (JPL)

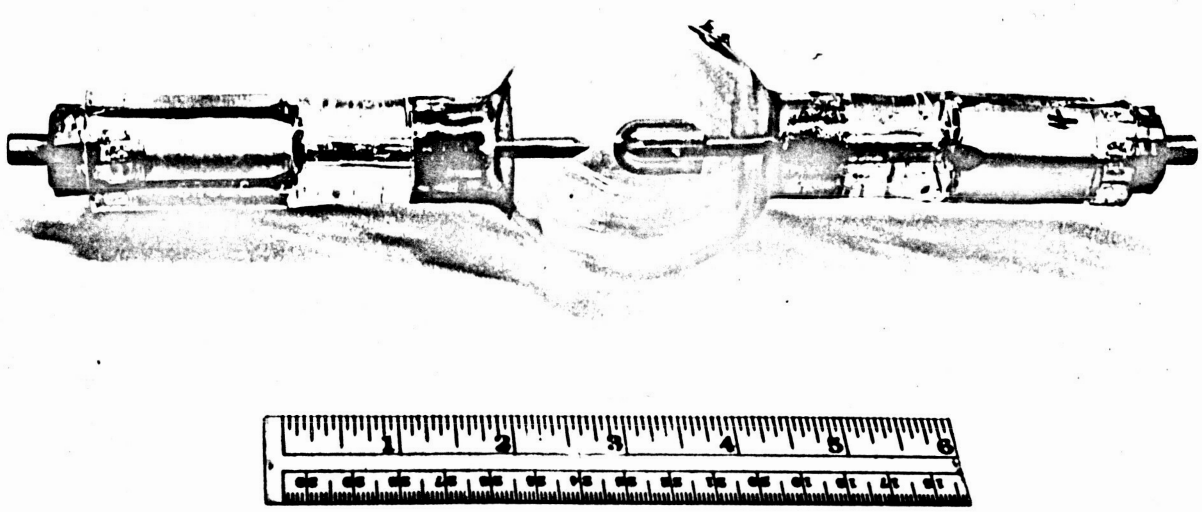


Fig. 6. Hanovia 2.5-KW mercury-xenon lamp

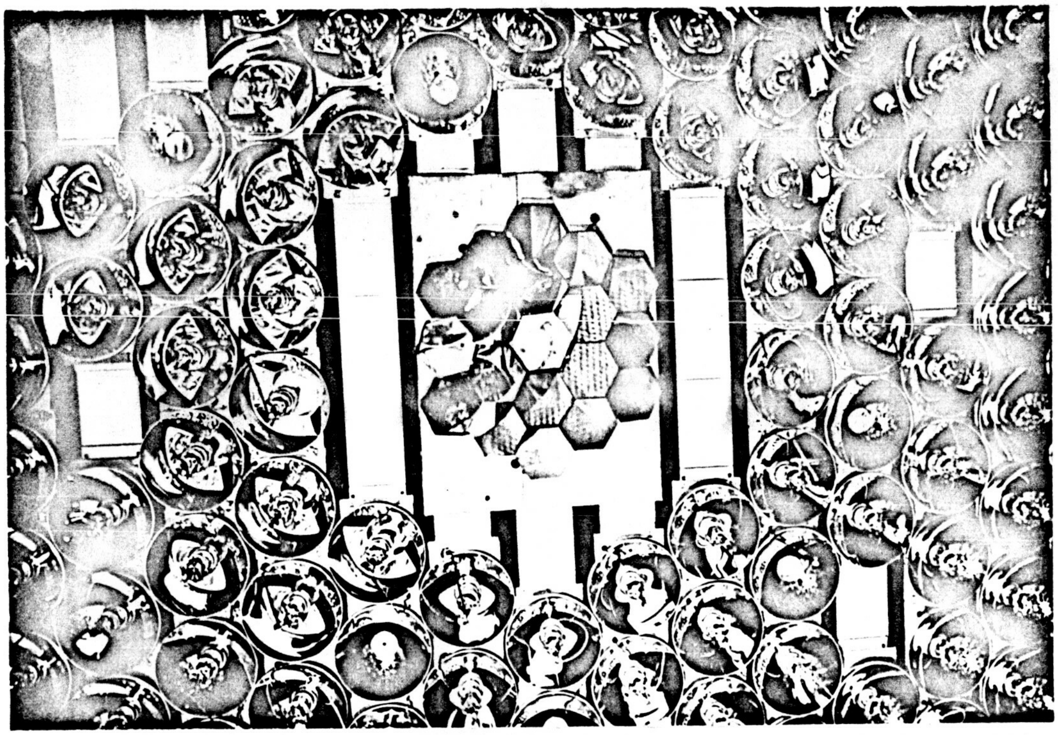


Fig. 7. Pseudo-hyperbola and lamp-reflector assemblies (JPL)

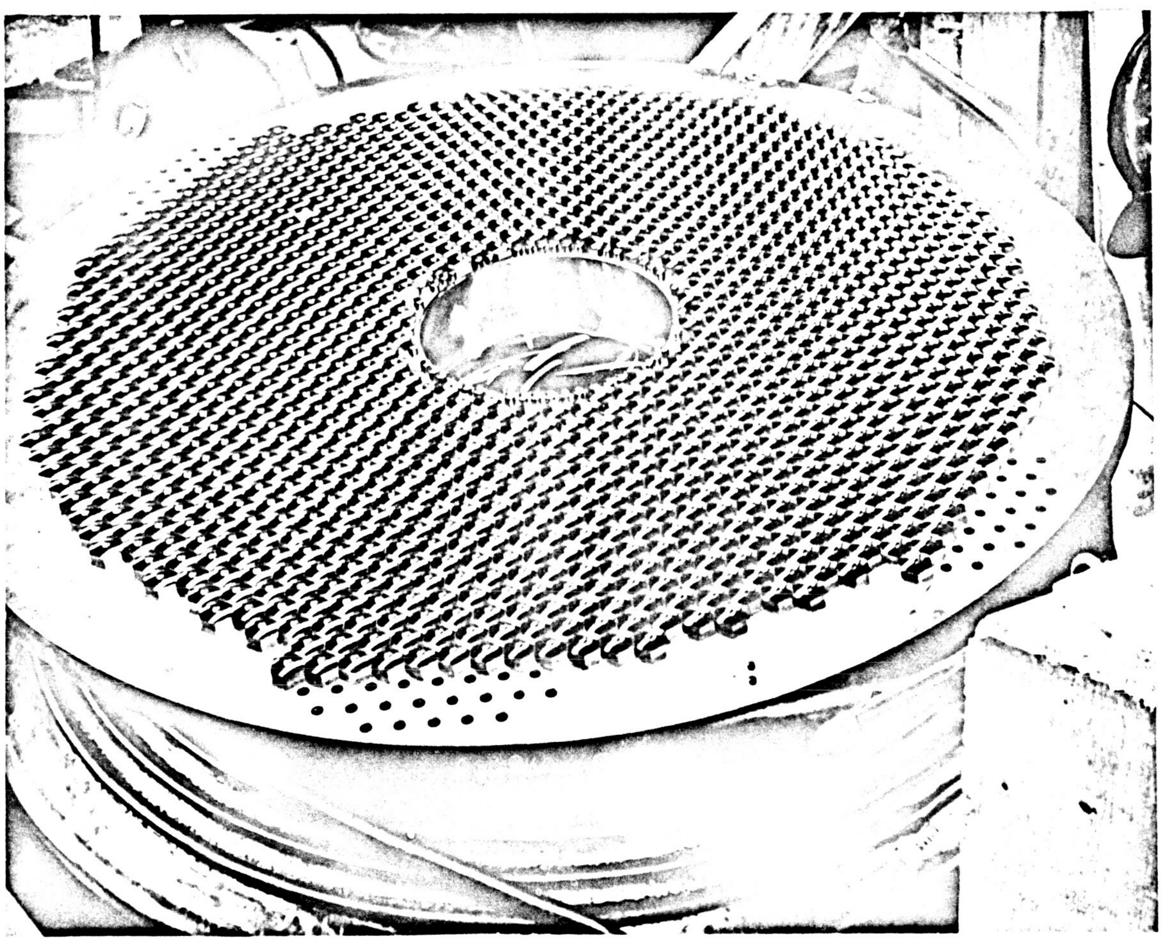


Fig. 8. Primary virtual source reflector (JPL)

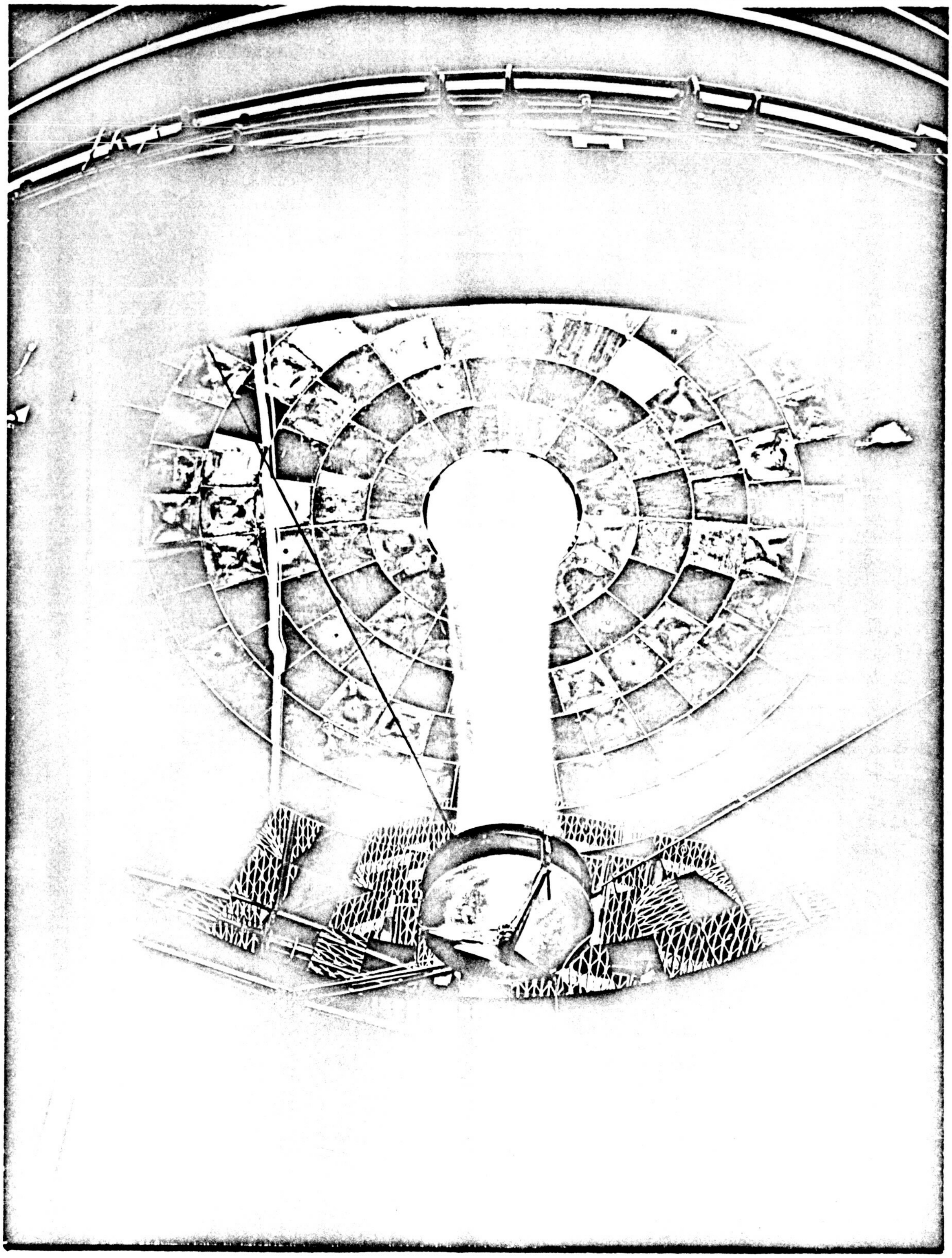


Fig. 9. Internal parabola, virtual source assembly, and transfer lens (JPL)

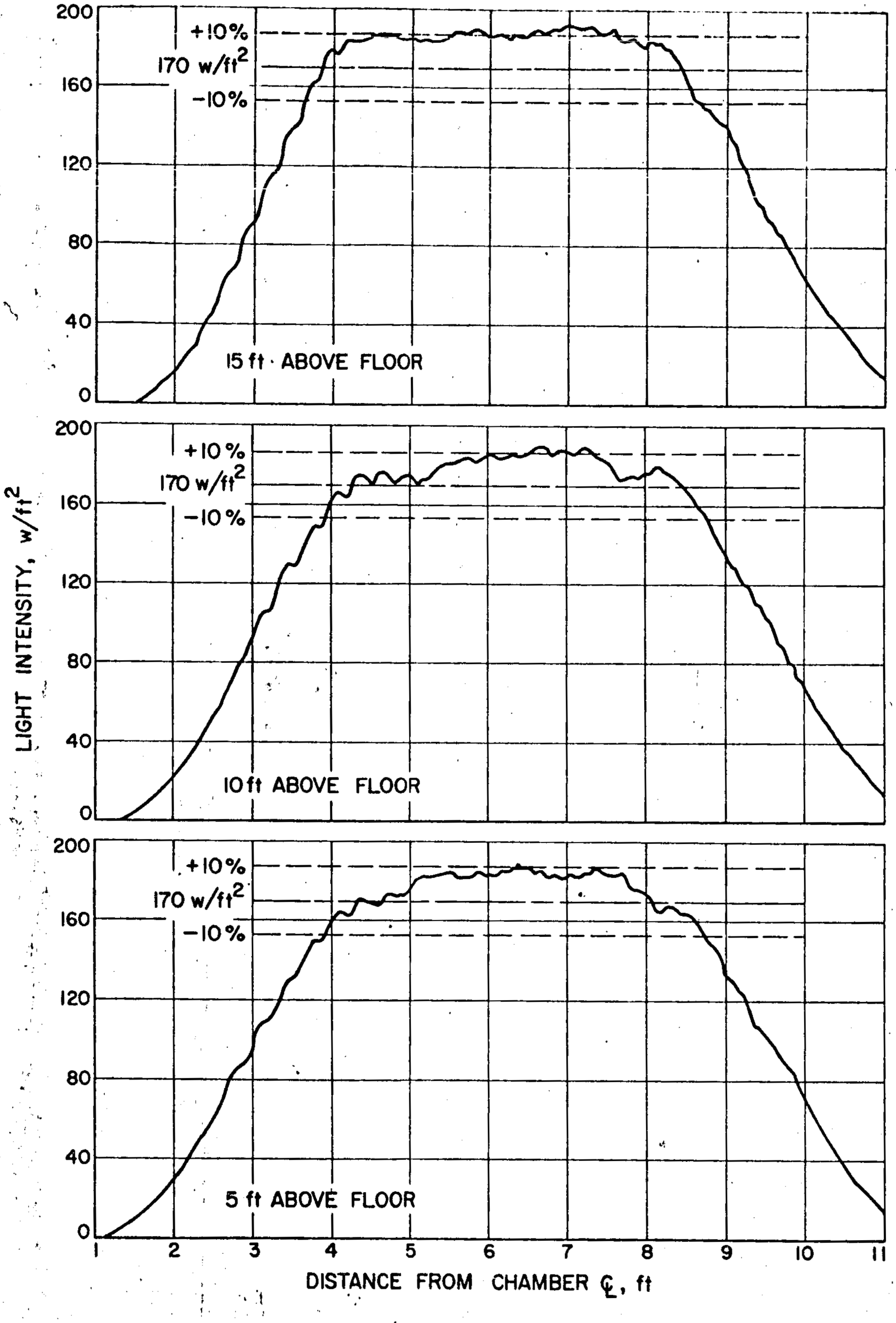


Fig. 10. Energy distribution across flats of hexagonal beam for 5-foot off-axis optical system (JPL)

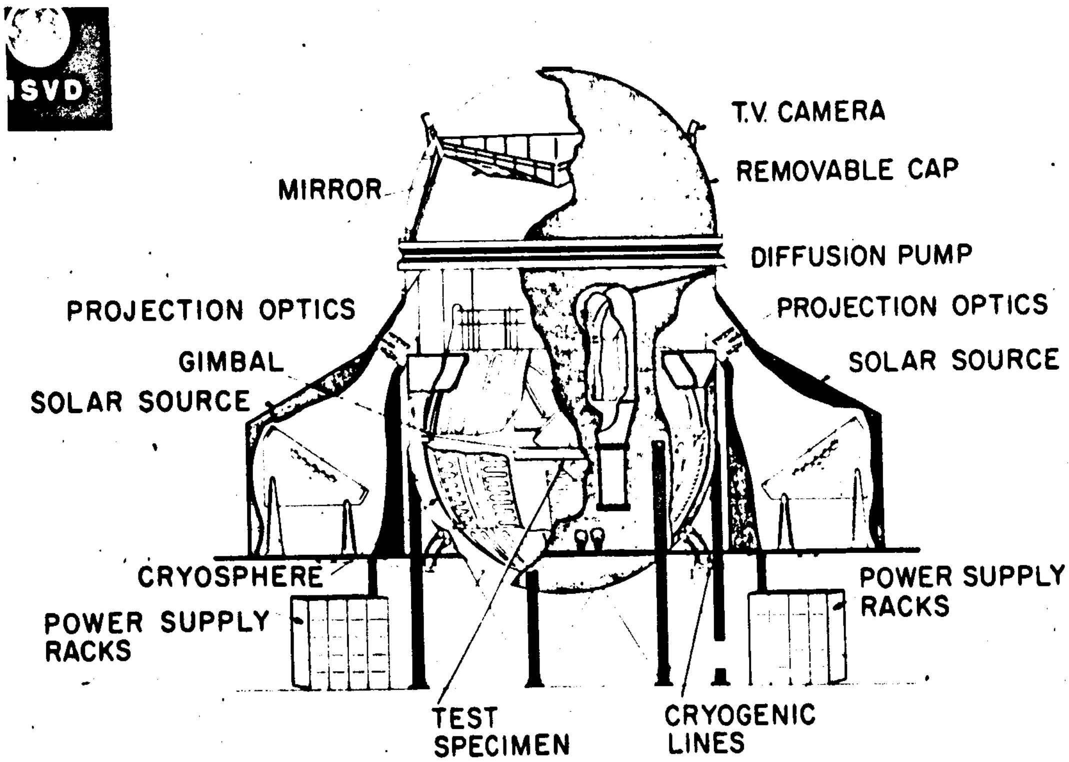


Fig. 11. Cutaway drawing of the General Electric Company's Space Environmental Simulation Laboratory

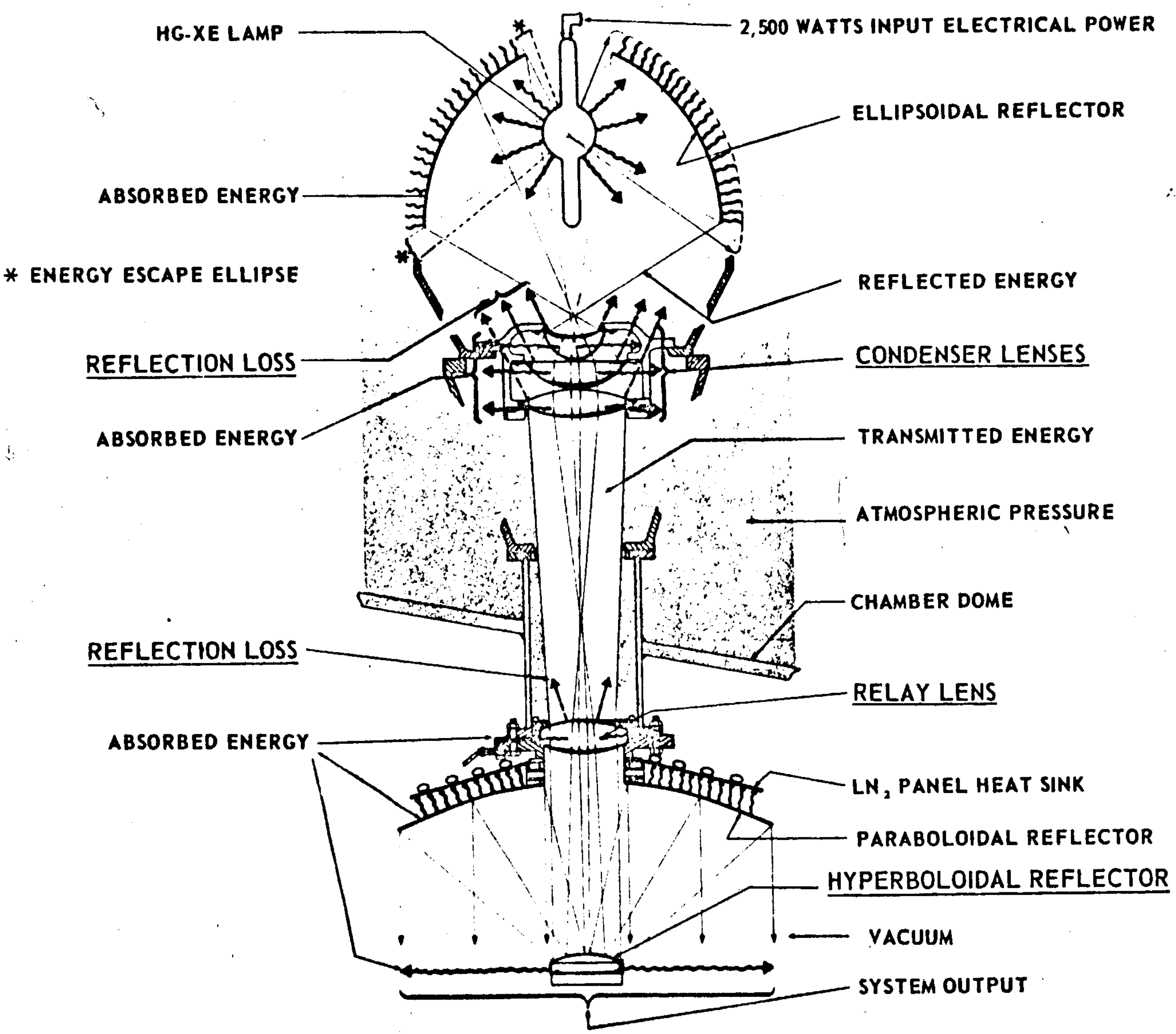
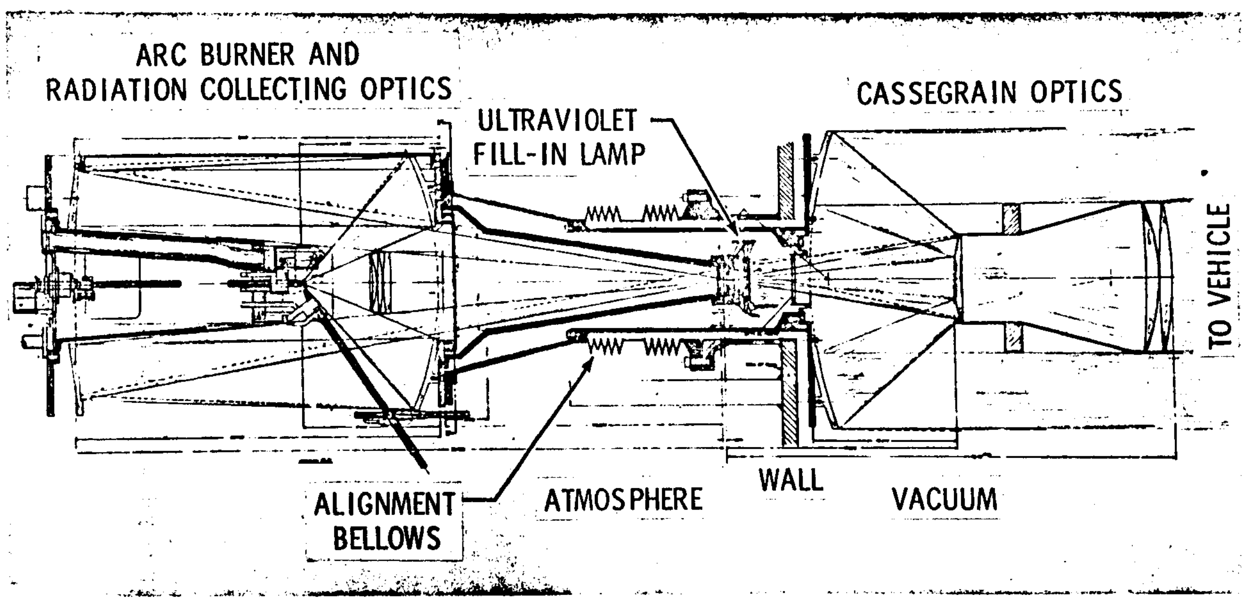


Fig. 12. Schematic drawing of solar simulator module for Goddard Space Flight Center



(Section View Less Magazine and Controls)

Fig. 13. Schematic drawing of solar simulator module for the Mark I facility at the AEDC

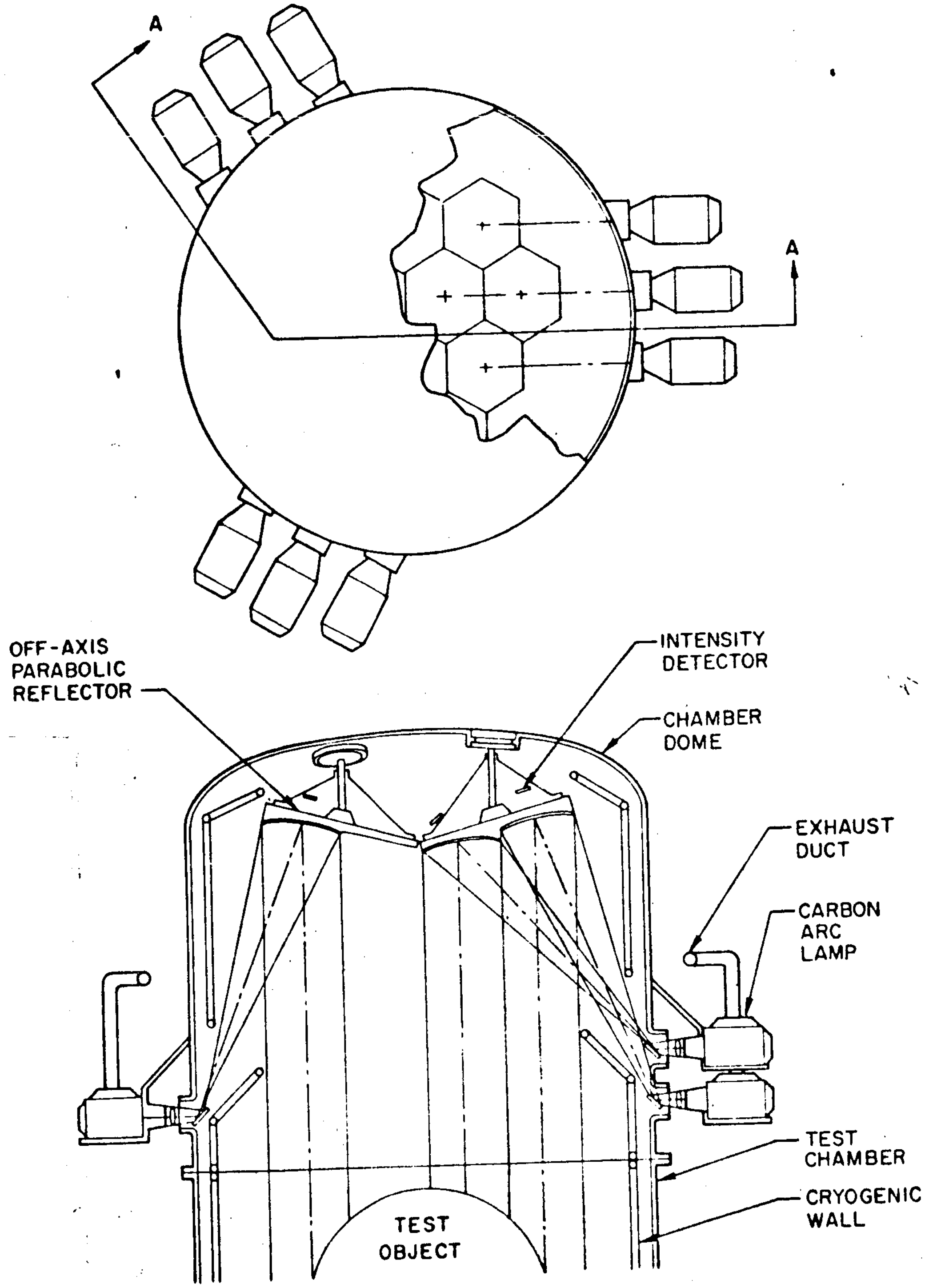


Fig. 14. Schematic drawing of off-axis, modular solar radiation simulation system proposed by Space Technology Laboratories

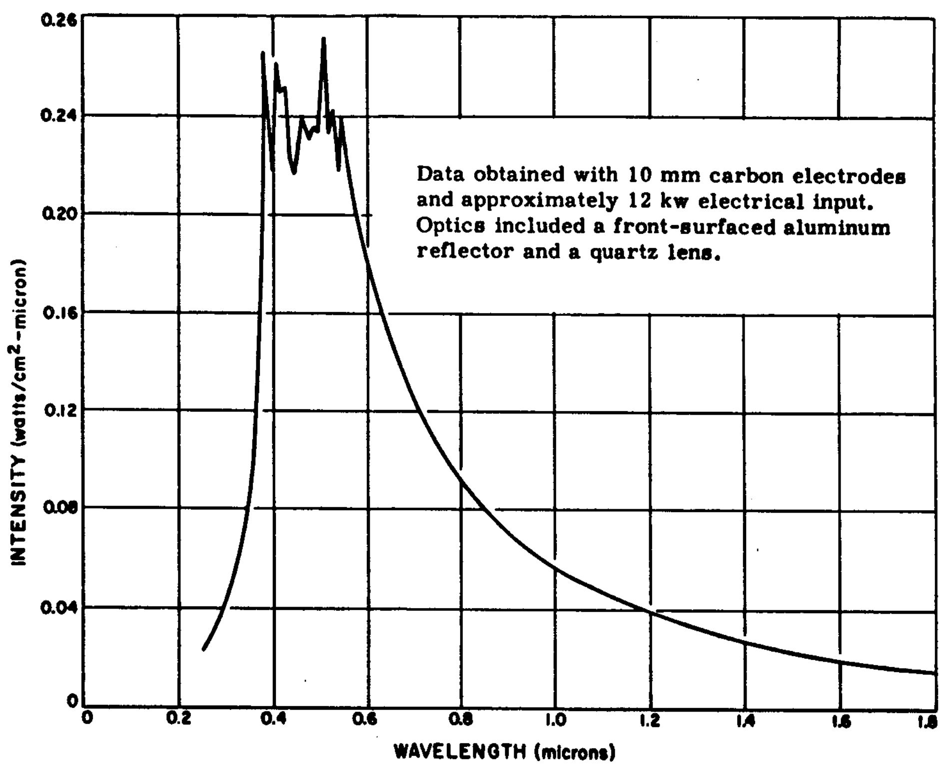


Fig. 15. Spectral energy distribution of a carbon-arc lamp

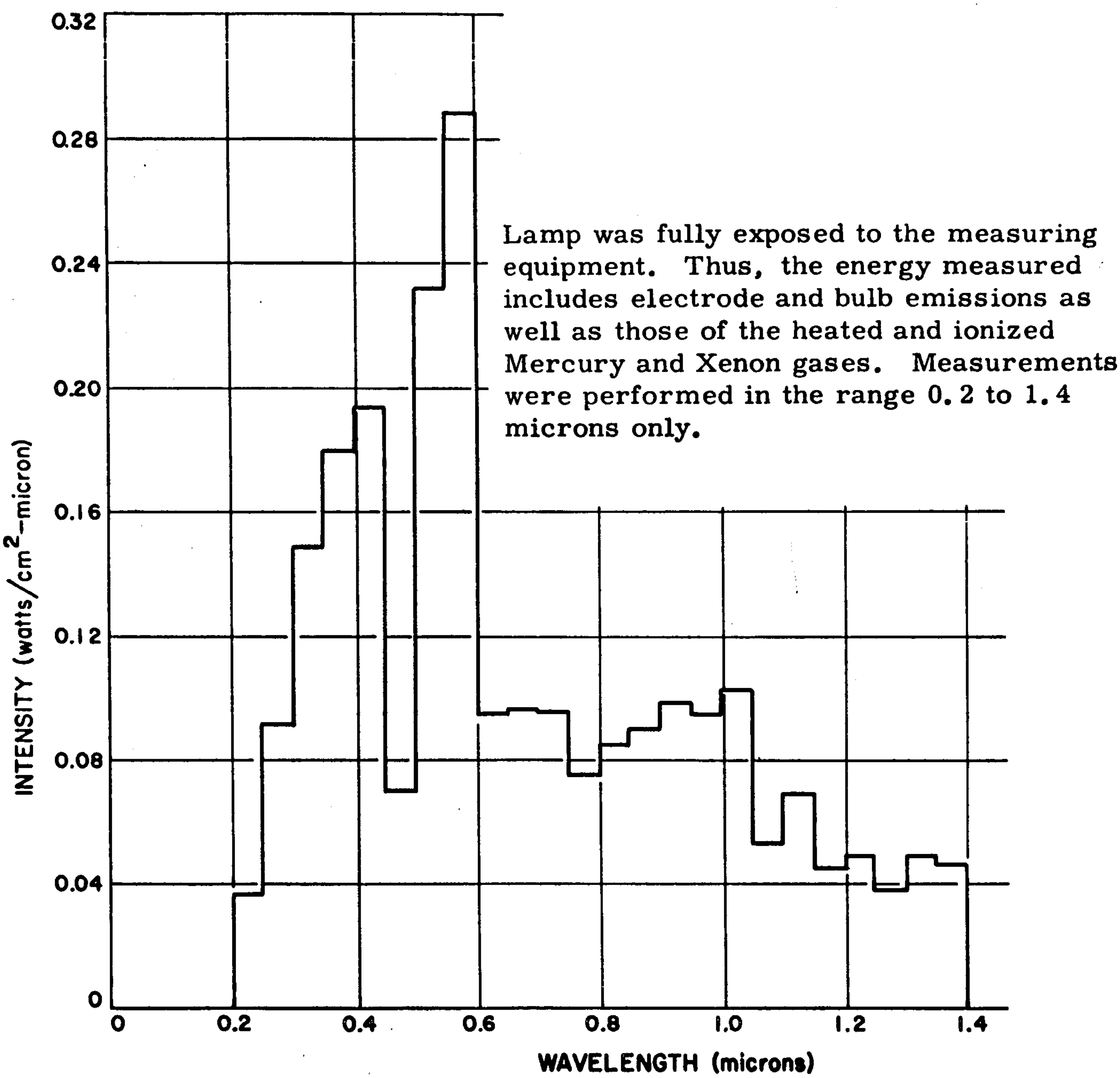
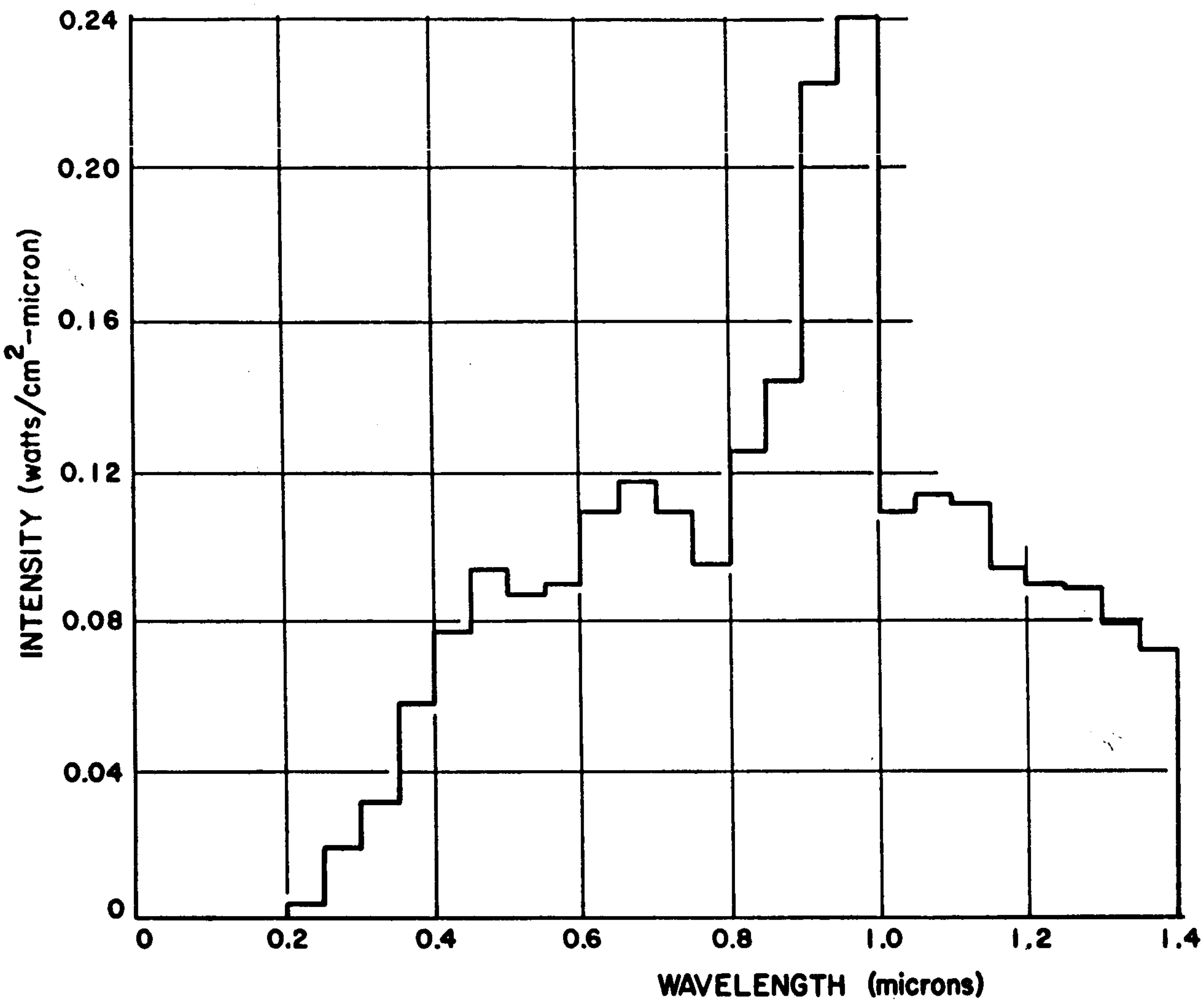


Fig. 16. Spectral energy distribution of a mercury-xenon compact-arc lamp



Lamp was fully exposed to the measuring equipment. Thus, the energy measured includes electrode and bulb emission as well as those of the heated and ionized Xenon gas. Measurements were performed in the range 0.2 to 1.4 microns only.

Fig. 17. Spectral energy distribution of a xenon compact-arc lamp

II. SIMULATION OF ALBEDO AND PLANET THERMAL RADIATION

The thermal energy originating at a body such as the Earth, Moon, or another planet constitutes an important input in the heat balance of a near satellite or other spacecraft. Two components of this energy can be identified. One of these, referred to as albedo, is that portion of the incident solar radiation which is reflected or scattered back into space by the planet's surface and atmosphere. Albedo radiation consists of ultraviolet, visible, and infra-red energy in about the same wavelength band as solar radiation, that is, from about 0.3 to 4 microns. The second component is referred to as "planet radiation" or "planet emission". This energy consists of the infra-red radiation associated with the body by virtue of its temperature, and thus appears in the wavelength range from about 4 or 5 microns out to 40 or 50 microns.

The intensity of these radiations varies widely. Typical values of albedo are about 0.35 for Earth, 0.10 for the Moon, 0.15 for Mars, and 0.70 for Venus, but actual values and the spectral distribution depend in detail upon the wavelength of the incoming radiation and upon surface conditions and cloud cover. Additional variables to be considered in estimating the energy reaching a spacecraft include the relative positions of the sun, planet, and spacecraft, and of course the distance of the spacecraft from the body in question. For typical satellites in orbits a few hundred miles above the Earth, the total intensity of the Earth-related radiation would be a few tenths of the direct solar intensity. Consequently, it is by no means negligible. In general, it can be said that albedo and planet emission constitute a significant source of energy for spacecraft closer than about one or two diameters of the reflecting body.

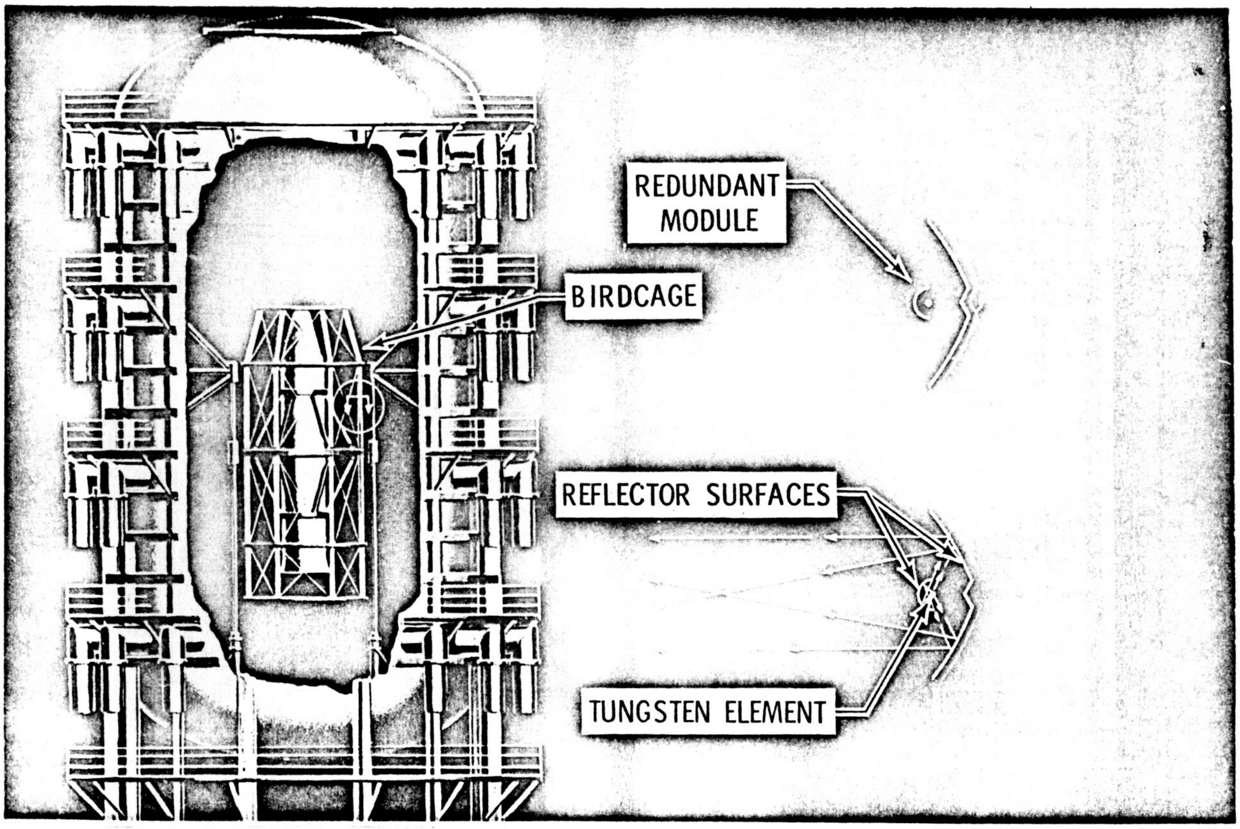
Considering all of the variables involved, it is clear that the requirements for simulation of this energy input will be peculiar to each occasion, that is, to each particular spacecraft and to its orbit or trajectory. For this reason, rather versatile simulation systems are required. The problem is rather different, however, from that of simulating solar radiation. The albedo and planet radiations are diffused rather than collimated, and the reflecting body will generally subtend a fairly large angle from the spacecraft (for example, about 135 degrees in a 300-mile Earth orbit). Thus, it would appear that many sources, or fewer rather extended sources, will be necessary to cover the required areas on the spacecraft. A portion of the albedo radiation is in the range where some materials are selective in absorptivity, but part of this energy and probably all of the planet infra-red radiation occurs in a range where materials are likely to have a flat spectral response. For these reasons, it is probably not necessary to be concerned about the wavelength of the aritificial radiation, but simply to expect that adequate simulation will be obtained if the proper energy flux is produced.

As an example, the arrangement proposed for the Mark I Aerospace Systems Environmental Chamber at the AEDC is shown in Figure 1 (from Reference 4). It consists of a "birdcage" composed of bare tungsten filaments and reflectors arranged in modules around the test item. The on-off time of each filament is programmed to provide the proper energy flux from each direction.

REFERENCE

1. Charles F. Norman, "Solar Simulation Instrumentation", Arnold Engineering Development Center, Report No. AEDC-TDR-62-191, January, 1963.

FIGURE



Albedo-Earthshine Simulator, Mark I

Fig. 1. Albedo and planet radiation simulator for the Mark I facility at the AEDC